

Spacecraft Design Thermal Control Subsystem

Robert N. Miyake
Jet Propulsion Laboratory
California Institute of Technology
Mission and Technology Development Group
Section 3542
31 January 2008

Agenda

- Introduction
- Thermal Control Techniques
- Heat Transfer Mechanisms
- Spacecraft Thermal Modeling and Analysis
- Thermal Environment
- Spacecraft Energy Balance
- Spacecraft Design Validation
- Spacecraft Design Reviews
- Spacecraft Mission Operations

Introduction

- The Thermal Control Subsystem engineers task is to maintain the temperature of all spacecraft components, subsystems, and the total flight system within specified limits for all flight modes from launch to end-of-mission.
- In some cases, specific stability and gradient temperature limits will be imposed on flight system elements.
- The Thermal Control Subsystem of “normal” flight systems, the mass, power, control, and sensing systems mass and power requirements are below 10% of the total flight system resources.
- In general the thermal control subsystem engineer is involved in all other flight subsystem designs.

Thermal Control Techniques

- The thermal control techniques are grouped as passive and active.
 - Passive Passive techniques are generally preferred because of simplicity, reliability, and lower risk and cost. Passive techniques are elements that once installed require no further spacecraft resources as control, electrical power, and does not impose any data handling requirements, does not change any property (such as heat rejection capability), has no external moving parts, and does not impose any data handling requirements.
 - Active Active techniques are used when passive techniques cannot provide the control needed. Active thermal control techniques use heating or cooling, thermal transfer, variable rejection, and sensing devices. Active techniques require spacecraft resources, including electrical power, data handling and control, and sensing and data storage. Active thermal control techniques require more testing, and have higher risk, and costs.

Passive

- Passive Thermal Control Flight Hardware

Multilayer Insulation (MLI)

Thermal Surfaces Paints / Films / Chemical Coatings

Thermal Conduction Control

Isolation Low Conductance Material (G-10)

High Conductance Al / Cu / Be / K1100 / APC

Thermal Storage
Mass of Materials
Phase Change Materials

Thermal Radiators

Normal Temperature

Cryogenic Passive Cryogenic

Thermal Transfer Heat Pipes

Radiation Block Sun Shade / Planet Shade

Entry System Heat Shield / Back Cover

Active

- Active Thermal Control Flight Hardware

- Coolers

- Sterling

- Sorption

- Thermal Transfer

- Heat Pipes

- Pumped Loops

- Thermal Louvers

- Thermal Switches

- Controllers

- Thermostats

- Heaters

- Electric

- Radioisotope

- Sensors

- Thermistors

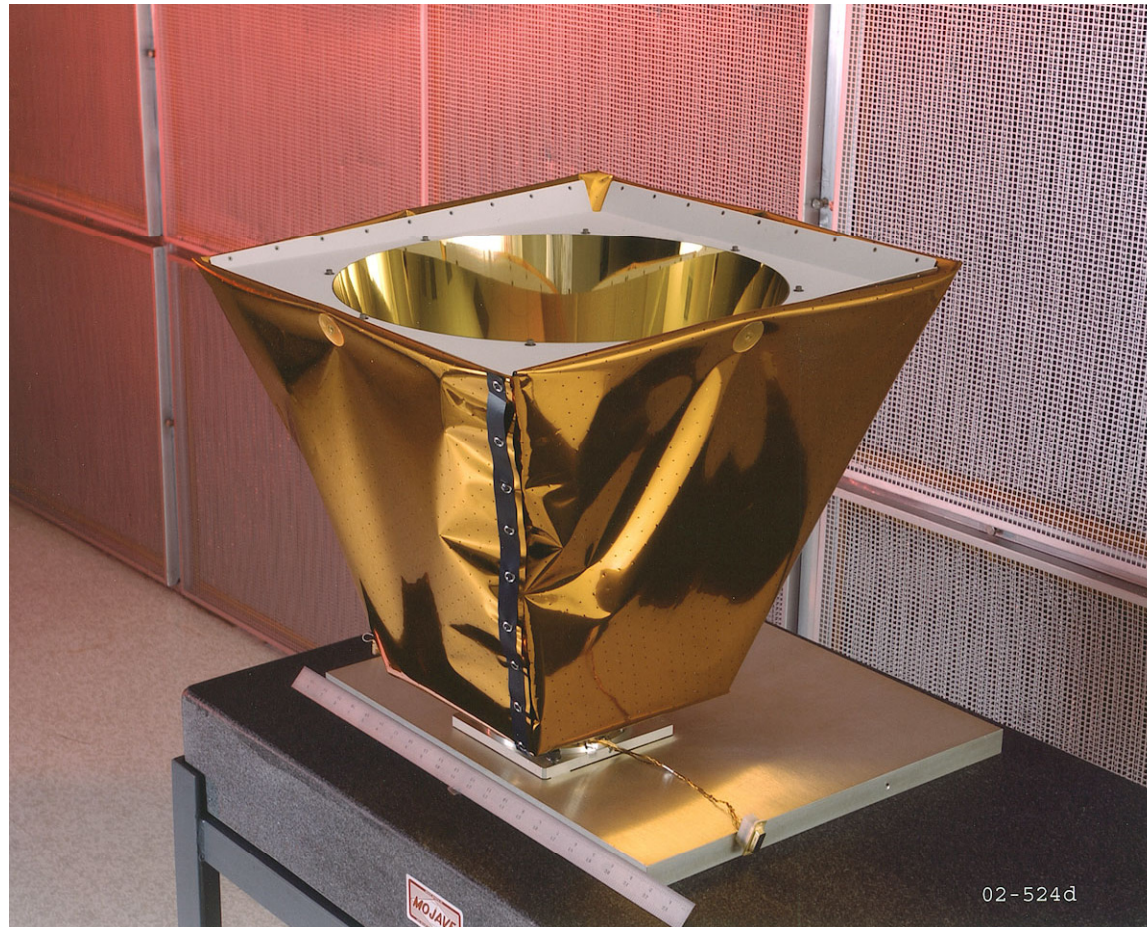
- PRT's (Platinum Resistance Thermometers)

- Thermocouples

- Dewars

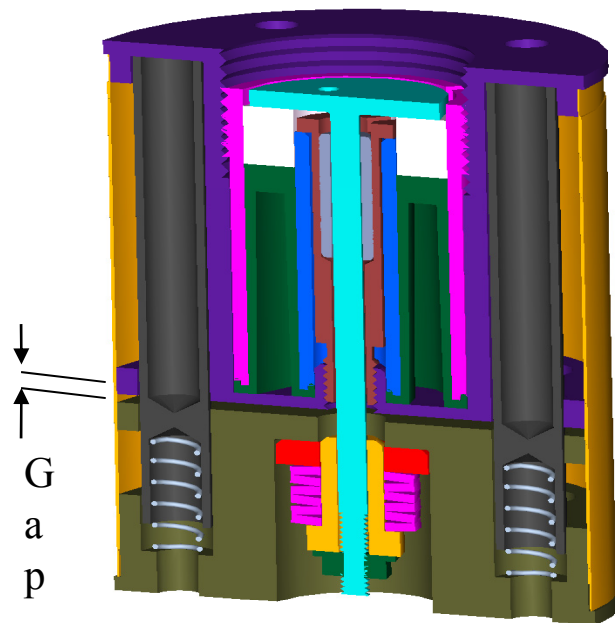
Passive Cryogenic Radiator

- Two Stage Passive Cryogenic Radiator

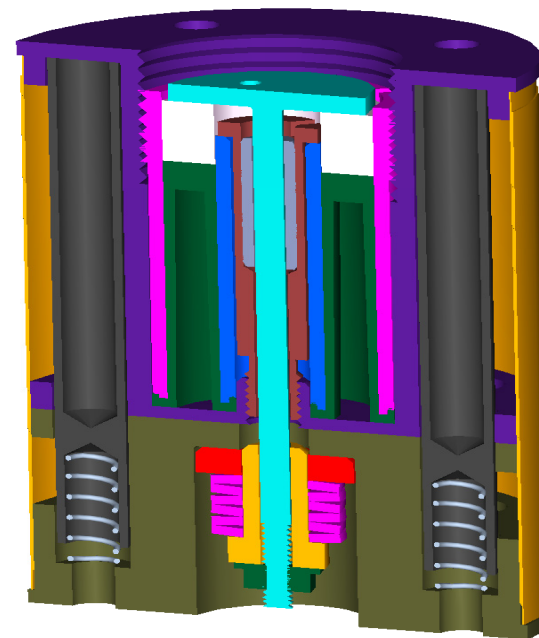


Thermal Switch

- Wax Pellet Thermal Switch



Switch Open



Switch Closed

Heat Transfer Mechanisms

- The heat transfer mechanisms affecting spacecraft thermal control are the same as that on Earth, which are conduction, convection, and radiation, although for unmanned spacecraft in space convection does not exist in space as there is no atmosphere or gravitational field. Even for manned spacecraft in space, since there is no gravitation there is no convection. Convection exists on planets with atmosphere such as Mars, Venus, Earth, and on moons with atmosphere such as Titan (a moon of Saturn).

Thermal Conductive Heat Transfer

- Thermal Conduction is thermal energy transfer occurs in the mechanical elements of the spacecraft. In general the structure utilizes metals such as aluminum, titanium, and steels, and composite materials an example being graphite epoxy.
- Thermal conduction design can be also used to minimize thermal transfer by material choice (example is G-10).
- The basic mathematical description for heat conduction is Fourier's law, the one dimensional form is:

$$Q = -kA(dT/dx)$$

Often energy flux is useful $q = Q/\text{unit area}$

Thus: $q = -k (dT/dx)$

and the conduction through a slab: $q = -k (T_L - T_O)/L$

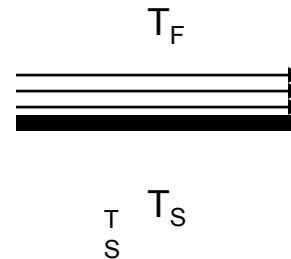
Where k = thermal conductivity; T_L = Temp at L; T_O = Temp at O; and
 L = length of slab.



Thermal Convective Heat Transfer

- Thermal convection is the transfer of thermal energy by the motion of material. An example is the transfer of thermal energy by movement of air on the Earth either by natural convection (by density differences, driven by gravity) and forced convection as the air is moved by fans.
- Convection - The mathematical description for convective heat flux is Newton's law of cooling (in one dimension):

$$Q = h_c A (T_s - T_F)$$



Where h_c = Convective heat transfer coefficient
 A = Heat transfer area

Thermal Radioactive Heat Transfer

- Thermal energy transfer by radiation is temperature difference driven. The only transfer of energy form from the spacecraft is by thermal radiation. (Note: For the Earth, the final energy dissipation in the energy balance is thermal radiation to space.).
- The Stephan-Boltzmann law states the power emitted by a body is:

$$Q = \varepsilon \sigma A T^4$$

The total emissive hemispherical power E is:

$$E = q = Q/A = \varepsilon \sigma A T^4$$

Where σ is the Stephan-Boltzmann constant $5.6704 \times 10^{-8} \text{ W/m}^2 \text{ K}^4$

ε is the emissivity of the surface.

Thermal Modeling and Analysis

- Thermal modeling and analysis utilizes thermal analysis tools. There are two classes of tools. The first class are the tools that develop the thermal radiation interchange between the several elements of the spacecraft. The second class are the analysis tools that use the spacecraft type of flight, thermal conduction network, external environmental inputs, and the internal energy dissipation to develop the thermal control design, determine thermal control subsystem hardware requirements, and flight temperature predictions and power requirements.
- Thermal radiation interchange tool – This tool develops the geometric factors that determine the radioactive thermal interchange.
- Thermal model analysis tools - There are two basic types of analysis tools, Finite Element Analysis and Finite Difference Analysis.
 - Finite Difference Analysis Tool utilizes direct solution of up to a 4-th order equations. This tool can directly analyze thermal systems.
 - Finite Element Analysis Tool utilizes a liner equation solution, with a simulation for the higher order equations. Several tools have been developed and their results are similar to that for the FDA.

Spacecraft Thermal Analysis Tools

Thermal Spacecraft Modeling Tools

Current thermal analysis tools are integrated and develops:

Thermal radiation interchange between flight elements and the environment.

Calculates the external environmental inputs. (from target information / Orbit)

Determines thermal conduction interaction between flight elements

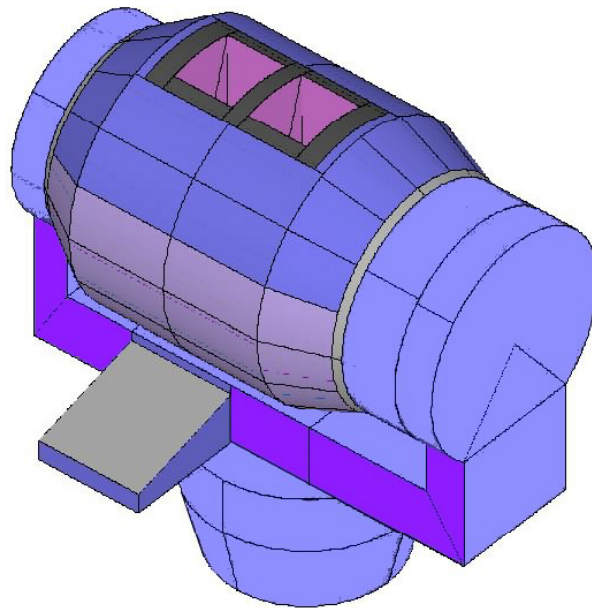
Develops the thermal control design, and determines electrical power required and predicts flight temperatures for all flight modes from launch to end-of-mission.

Thermal analysis tools used are:

CINDA-SINDA / TSS / TMG / Thermal Desktop

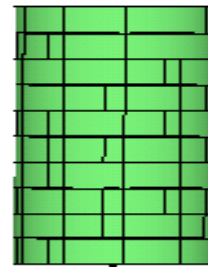
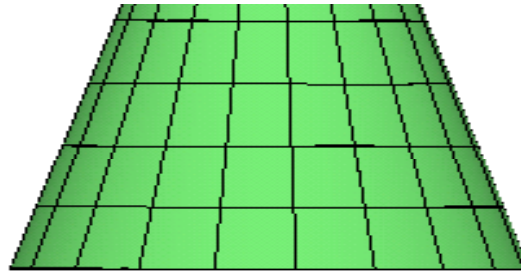
Instrument Thermal Model

- Typical Instrument Math Model – Nodes



Structure Thermal Nodes

Typical Structure Math Model Nodes



External Thermal Environment

- The External Environment inputs are due to thermal interchange from sources external to the spacecraft. The following are the external environmental sources.

Solar Incident	This is incident energy from the Sun and varies as the distance the flight system is from the Sun.
----------------	--

Body Reflected	This reflected solar energy from the body surface. The intensity on the spacecraft depends on the albedo and altitude above the body.
----------------	---

Body IR	This is the energy radiated from the body which the spacecraft is orbiting or during fly-by.
---------	--

Other	One example is the Universe thermal background (~2 to 3 K).
-------	---

External Environment - Solar System Data

Solar and Planetary Data											
Planet/Satellite	Radius (km)	Distance to Sun (AU)			Flux (W/m ²) Mean	Rot. Period (Earth Days)	Length of Year (Earth Days)	Temperature (K)			Albedo
		Perhelion	Mean	Apohelion				Day	Avg	Night	
SUN	696100								5780		
Mercury	2439.7	0.308	0.387	0.467	9126.00	58.646	88.00	773	440	326	0.106
Venus	6051.9	0.720	0.723	0.728	2613.70	-243.010	225.00		737		0.65
Earth	6378.14	0.985	1.000	1.020	1367.50	0.997	365.25	295	288	275	0.367
Earth Moon	1738				1376.50	-27.322		392	253	100	0.12
Mars	3397	1.380	1.524	1.660	589.00	1.026	687.00	300	208	170	0.15
Jupiter	71492	4.950	5.203	5.460	50.52	0.414	11.86 yrs		163		0.52
Saturn	60268	9.040	9.539	10.120	15.03	9.438	29.46 yrs		133		0.47
Uranus	25559	18.320	19.191	20.080	3.71	-0.650	84.07 yrs		78		0.51
Neptune	24764	29.910	30.061	30.390	1.51	0.768	164.80 yrs		73		0.41
Pluto	1151	29.650	39.529	48.830	0.87	-6.387	248.50 yrs		48		0.3

NOTE: All planet orbits are within 3.5 degrees of the plane of the ecliptic except Mercury (7 degrees) and Pluto (17.1 degrees).
(The Earth Moon orbit is 5.1 degrees.)

External Environment – Simple Calculations

- To determine the magnitude of the external environmental incident energy a simple calculation can be made.
- For a mission within the Solar System (In reality all current missions are within the Solar System.)
 - The Major source of energy is the Sun.
 - A secondary source of energy are near objects, such as planets or moons.
- To evaluate the thermal energy source on the flight system for early evaluation can be calculated as follows (using the nominal Earth range of Solar incident energy of 1367.5 Watts/m²):

Direct Solar Incident Energy

$$\text{Incident Energy} = 1367.5 \text{ Watts/m}^2 / (\text{AU range of the Flight System})^2$$

Example: Direct Solar Incident Energy at Venus and Mars.

$$\text{Venus (Range from Sun} = 0.72 \text{ AU)} = 1367.5 / 0.72^2 = 2637.9 \text{ Watts/m}^2$$

$$\text{Mars (Range from Sun} = 1.52 \text{ AU)} = 1367.5 / 1.52^2 = 591.9 \text{ Watts/m}^2$$

Thermal Balance – Flight System Temperature Estimate

- Using the incident energy developed in the previous calculation, an estimate of a flight system bulk temperature can be determined.
- For a quick evaluation the following calculations can be done:
- Determine the range from the Sun in AU.
- Assume the flight system is a sphere (A cube can be used, but the calculation is more complex, although the cube calculation will allow for variation in thermal control surfaces.).
- Assume thermal optical properties (Initial calculations can be made for a black surface or white surface.).
- Add the internal electric power dissipation and any other thermal energy source (e.g. RHU's).

Spacecraft Bulk Temperature Calculation

- Simple Example
- Assume The Spacecraft is a Black Sphere ($A_s/E = 0.8/0.8$)
 Range from Sun 0.72 AU (Venus distance) and 1.52 AU (Mars distance)

- Calculate Bulk Temperature

At 0.72 AU

$$\text{Range from Sun} = 0.72 \text{ AU} = 1367.5 / 0.72^2 = 2637.9 \text{ Watts/m}^2$$

$$Q_{\text{in}} = Q_{\text{out}}$$

$$Q_{\text{in}} = \text{Incident Direct Solar} \times \text{Projected Area} \times \text{Solar Absorptivity}$$

$$Q_{\text{out}} = \text{Stephen-Boltzmann Constant} \times \text{IR Emissivity} \times \text{Total Area} \times (\text{Temperature})^4$$

$$\text{Temperature} = [(2637.9 \times 1 \times 0.8) / (5.6704 \text{ E}^{-8} \times 0.8 \times 4)]^{0.25} = 55.24^\circ \text{ C}$$

At 1.52 AU

$$\text{Range from Sun} = 1.52 \text{ AU} = 1367.5 / 1.52^2 = 591.9 \text{ Watts/m}^2$$

$$Q_{\text{in}} = Q_{\text{out}}$$

$$Q_{\text{in}} = \text{Incident Direct Solar} \times \text{Projected Area} \times \text{Solar Absorptivity}$$

$$Q_{\text{out}} = \text{Stephen-Boltzmann Constant} \times \text{IR Emissivity} \times \text{Total Area} \times (\text{Temperature})^4$$

$$\text{Temperature} = [(591.9 \times 1 \times 0.8) / (5.6704 \text{ E}^{-8} \times 0.8 \times 4)]^{0.25} = -47.13^\circ \text{ C}$$

- ABC

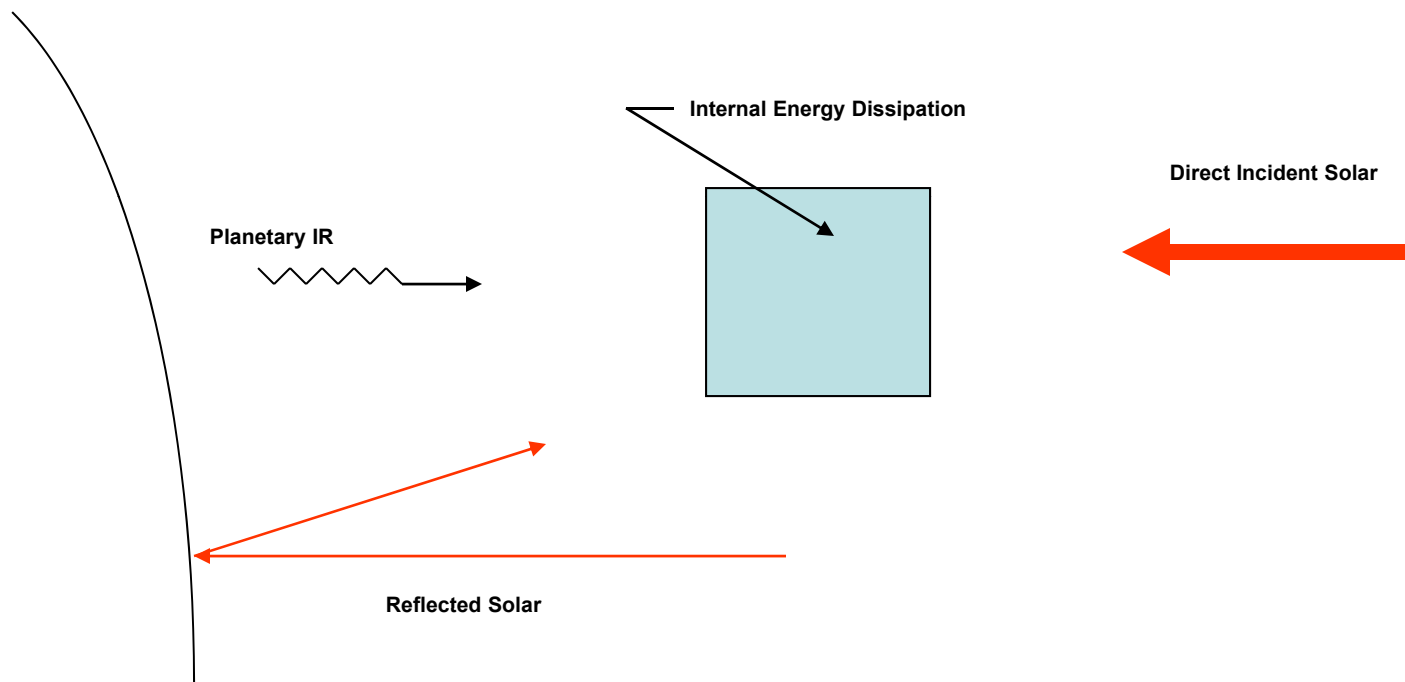
Flight System Driven Environment

- The Spacecraft must dissipate the total thermal load that is expected to be generated within the spacecraft structure. The thermal load consists of the electrical energy generated by the EPS and any other thermal source (one example are RHU's).
 - Internal Energy Dissipation – The internal energy dissipation is on the order of the electrical energy generated by the EPS (Electrical Power Subsystem). The total electrical dissipation is used in spacecraft thermal subsystem design because the electrical generation must be dissipated. This may be by the spacecraft surface or with thermal radiators.
 - Thermal Sources – For spacecraft the have radioactive elements, the thermal energy generation is a product of either electrical power generation (RTG's and Reactor's) or a thermal energy source (RHU's)

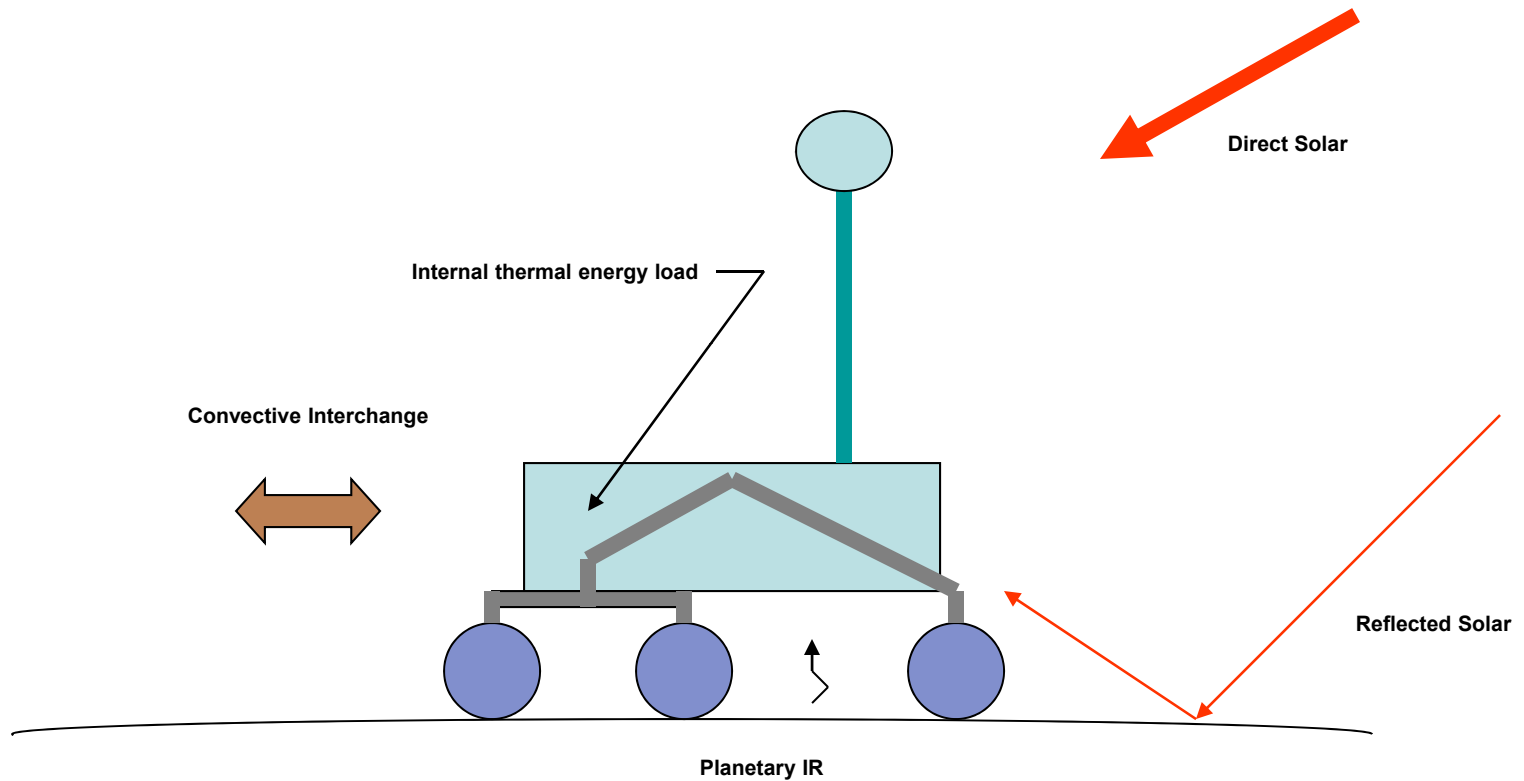
Spacecraft Energy Balance

- The spacecraft energy balance is the temperature the analysis determines from the total energy that the spacecraft absorbs Vs the energy rejected.
- As previously stated the absorbs is from the direct solar incident, reflected solar incident, planetary incident energy, and the internally generated thermal energy (which consists of generated electrical energy and other sources).
- For bodies with an atmosphere there are thermal convection energy transfer, which is determined by the body atmosphere gasses, temperature, and body gravity.
- For spacecraft with cryogenic elements such as detectors, optics, electronics, etc. The universe background temperature reduces the thermal rejection capability.

Spacecraft Energy Balance - Orbiter



Spacecraft Energy Balance - Lander/Rover



Spacecraft Design Validation

- Spacecraft design validation is done by testing. Spacecraft validation testing is done in a boundary control thermal vacuum chamber.
- This testing will be done on components, subsystems, and the spacecraft to verify the design, and analysis temperature predictions.
 - Component thermal testing in some cases can be done under normal atmosphere, if simple maximum / minimum operational and survival temperature range is to be verified.
 - For subsystem and system testing, thermal-vacuum testing is usually necessary.
 - For both subsystem and system testing the thermal math model is modified to present the test chamber boundary conditions, and temperature predictions for the test are made.
 - The test and predictions temperatures are compared and the differences noted will be used to modify the math model so that the predictions are the same as the test data. The changes are actual properties that are some time difficult to predict, such as joint thermal conductance.

Thermal Vacuum Test - Cassini Spacecraft



Spacecraft Design Reviews

- **There are several level of reviews which provide steps to determine the maturity of the design.**
- **Concept Design Review**
- **Preliminary Design Review**
- **Detail Design Review**

Conceptual Design Review

- **The Conceptual Design Review is done at the proposal input stage, to determine if the basic thermal control design will meet the science return mission.**
- **Develop basic design to determine if the spacecraft thermal control can be maintained within specified temperature level with the resources available and an initial hardware requirement.**

Preliminary Design Review

- **The Preliminary Design Review occurs prior to the detail design effort and describes the design at the end of the preliminary design.**
- **Provide initial design, with hardware requirements, electrical power and sensor requirements, and an initial mass and cost estimate.**
- **Determine if there are any procurement that is schedule sensitive, and develop procurement schedule.**
- **The total cost of the thermal control system, including all analysis and design costs, hardware costs including acquisition cost, testing cost both workforce and hardware, any development cost, and mission operations analysis and workforce cost.**

Detail Design Review

- **The Detail Design Review occurs at the end of the detail design process.**
- **Provides the detail design, with a detail description of thermal control subsystem hardware elements including the number and size of all hardware and the mass of the thermal control subsystem. Further provide a detail electrical power requirement for all flight modes, and the number and sensitivity of sensors. The total cost of the thermal control system is updated to meet the detail design hardware and workforce developed for the final detail design.**
- **Successful passing of the Detail Design Review is the go-ahead for purchasing hardware and beginning of subsystem fabrication.**

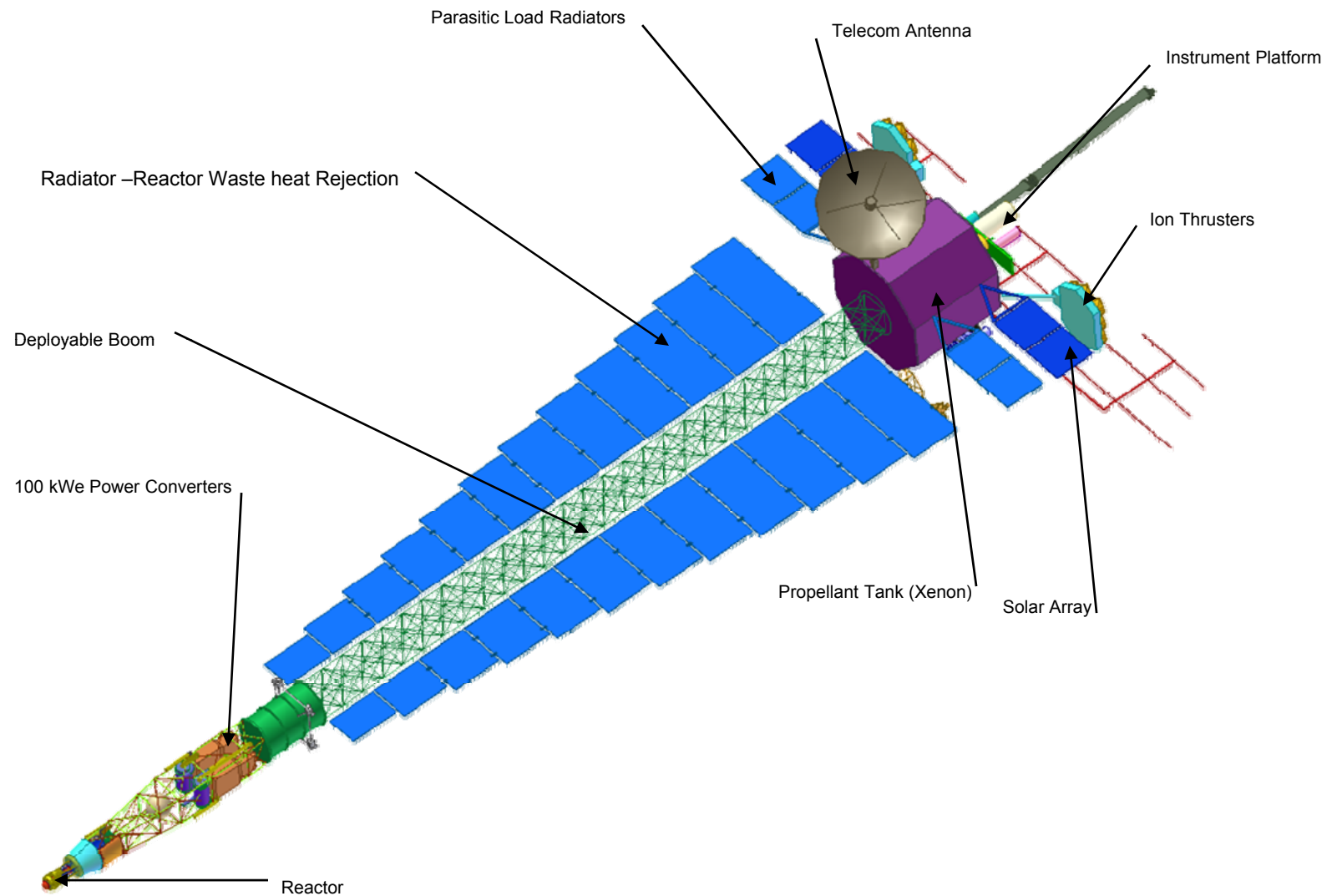
Spacecraft Mission Operations

- The thermal control subsystem is one of the subsystems that provide support during the flight operations from launch pad operations to end-of-mission.
- The process that the support is provided are as follows:
 - Utilize the development thermal math model as the basis for mission operations support.
 - This model is initially updated with data from the thermal vacuum testing to develop updated flight predictions.
 - During initial flight operations, the flight data is used to update the thermal math models and to provide mission temperature predictions.
 - The updated thermal math model is used to support anomalous flight conditions.

Spacecraft Thermal Control - Example

Example: Study	JIMO This is a mission to Jupiter to Study the Moons of Jupiter
Special Elements	This mission is nuclear powered electrical system with an electric power generating system at 200 kW e level (~1 mega W Thermal).
Reason	High Power Requirements for Propulsion, Telecom, Science.
Mission Design Requirements	
Target Planet	Jupiter
Other Targets	3 moons of Jupiter –Callisto, Ganymede, and Europa
Mission Life	up to 12 years
Environmental Effects	
Distance to Sun	5.2 AU - Solar incident energy 1/27 at Earth
High Radiation	Jupiter Environment
Induced Environment	Nuclear Reactor Radiation High reactor radiator temperature level (795 K)

Spacecraft Thermal Control – Configuration



Spacecraft Thermal Control – Thermal Block Diagram

